

**A Rocket Powered  
Single-Stage-to-Orbit Launch Vehicle  
With U.S. and Soviet Engineers**

**by**

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## SUMMARY

A single-stage-to-orbit launch vehicle is used to assess the applicability of Soviet Energia high-pressure-hydrocarbon engine to advanced U.S. manned space transportation systems. Two of the Soviet engines are used with three Space Shuttle Main Engines.

When applied to a baseline vehicle that utilized advanced hydrocarbon engines, the higher weight of the Soviet engines resulted in a 20 percent loss of payload capability and necessitated a change in the crew compartment size and location from mid-body to forebody in order to balance the vehicle. Various combinations of Soviet and Shuttle engines were evaluated for comparison purposes, including an all hydrogen system using all Space Shuttle Main Engines. Operational aspects of the baseline vehicle are also discussed. A new mass properties program entitled Weights and Moments of Inertia (WAMI) is used in the study.

## INTRODUCTION

A single-stage-to-orbit launch vehicle from reference 1 has been used to assess the performance and integrational aspects of a high-pressure hydrocarbon engine used on the Energia Booster (Ref. 2). The baseline configuration for the present study is one in which 3 Space Shuttle Main Engines (SSMEs) are used in conjunction with the 2 Soviet Engines designated RD-170 (Fig. 1). In the

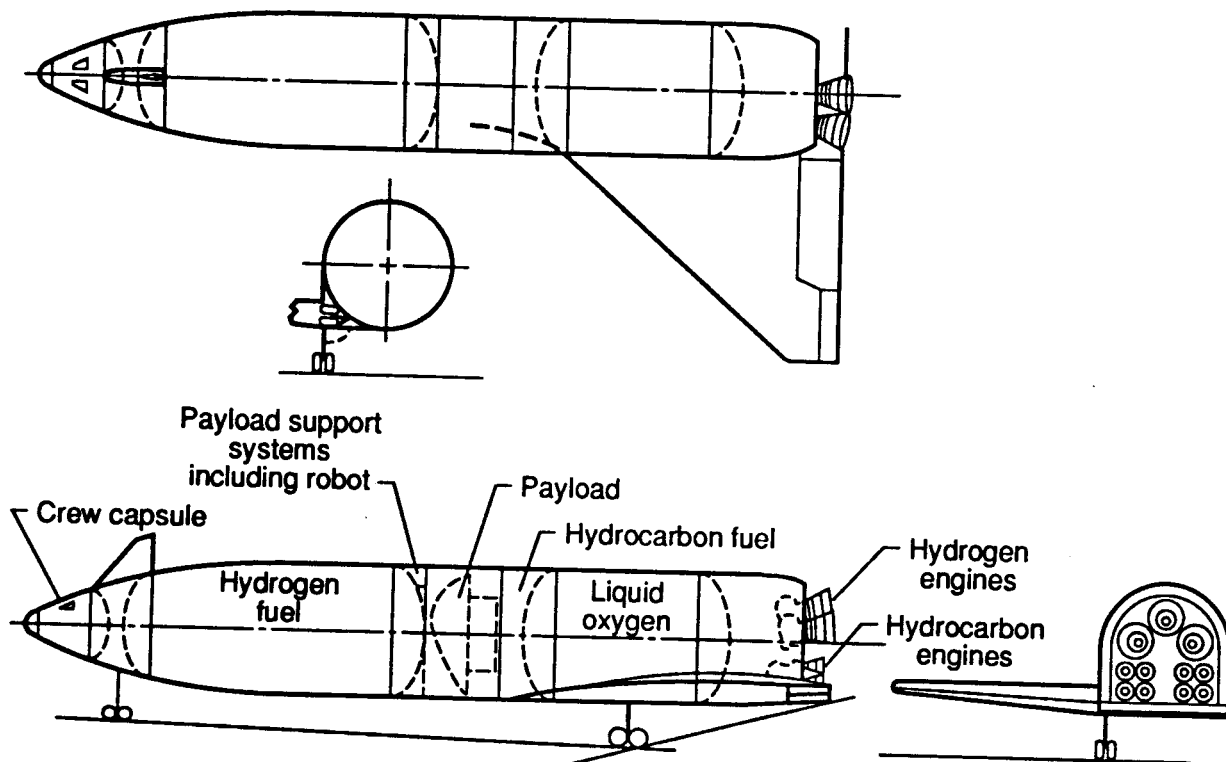


Figure 1. Dual-fuel single stage, inboard profile.

earlier study (Ref. 1), the hydrocarbon engines used were those described in reference 3. Because existing engines are used, the allocations for the weights of the SSME and RD-170 engines in the weights analyses are fixed quantities reflecting actual engines, while vehicle size and all other subsystems are allowed to vary in weight and size during optimization.

This study was undertaken in the continuing efforts to evaluate and better understand the various options for advanced transportation systems.

### **THE SOVIET ENERGIA BOOSTER ENGINE (RD-170)**

The Soviet RD-170 engine is the world's (highest thrust) hydrocarbon engine in use. Each engine consists of 1 turbopump assembly driven by 2 preburners which feed 4 thrust chamber assemblies. The engine uses a staged combustion power cycle and has been flown 29 times. The sea level static thrust and engine weights are almost identical to the Saturn F-1 engine giving almost identical sea level-delivered thrust-per-pound of engine (Table I). However, the chamber pressure for the RD-170 is over three and one half times that of the F-1, and the vacuum specific impulse is 337 seconds versus 304 seconds for the F-1.

Table I. Hydrocarbon Engine Comparison

Item	F-1	RD 170	Aerojet* Study
Engine dry weight, lb	18,616	19,305	4066
Sea level thrust, Mlb	1,522,000	1,631,000	530,000
Thrust-to-weight	81.8	84.5	130
Chamber pressure, psia	1122	3556	4000
Vacuum specific impulse, sec	304	336	350
Sea level specific impulse, sec	265	308	323
Expansion ratio, E	16	36	40

\*Engine proportionately downsized from NASA CR-135141 of Reference 2 and used in vehicle described in Reference 1.

### **SPACE SHUTTLE MAIN ENGINES (SSME)**

The SSMEs used in the study are assumed to be modified from the 77.5 expansion ratio to employ a dual-position nozzle with expansion ratios of 40 and 120 (Table II). All of the SSME engines were assumed to be operated at 109 percent of the current normal power level. The RD-170s were assumed to operate at the nominal published thrust values.

Table II. Hydrogen Engine Performance

Item	SSME E = 77.5			SSME E = 40/120		
	Normal Power Level	Emergency Power Level (109%)		Normal Power Level	Emergency Power Level (109%)	
Engine dry weight, lb	———	7500	———	———	8023	———
Sea level thrust, Mlb	373,500		415,600	413,693		455,343
Sea level thrust-to-weight	50		55	52		57
Chamber pressure, psia	3022		3277	3022		3277
Vacuum specific impulse, sec	454		454	458		458
Sea level specific impulse, sec	362		369	400		404
Expansion ratio, E	———	77.5	———	———	40/120	———

One vehicle design was studied with the original 77.5 expansion ratio SSME engines. These runs were made for the 3-SSME/2-RD-170 combination in order to compare vehicle weights for the better performing but heavier SSMEs with the dual-position nozzles.

### PROPULSION-VEHICLE SYSTEM CONFIGURATION CHARACTERISTICS

The primary propulsion-vehicle system configuration is that shown in Figures 1 and 2. The engine combination is sufficiently compact so as to permit some boattailing of the vehicle body, allowing for a 20 percent reduction in base area for a projected improvement in subsonic L/D (See configuration A).

The engine arrangement on the base of the vehicle for 6 SSMEs and 1 RD-170 is shown in figure 2B and for the 9 and 11 SSME combinations in 2C and 2D. The all-SSME propulsion systems with extendible nozzles tend to be crowded on the base and, in fact, the 11-engine design would require fairings that would extend beyond the base to protect the engines. Characteristics of the various engine combinations are tabulated in Table III. These constants are inputs for the weights program for the engines and thrust structure.

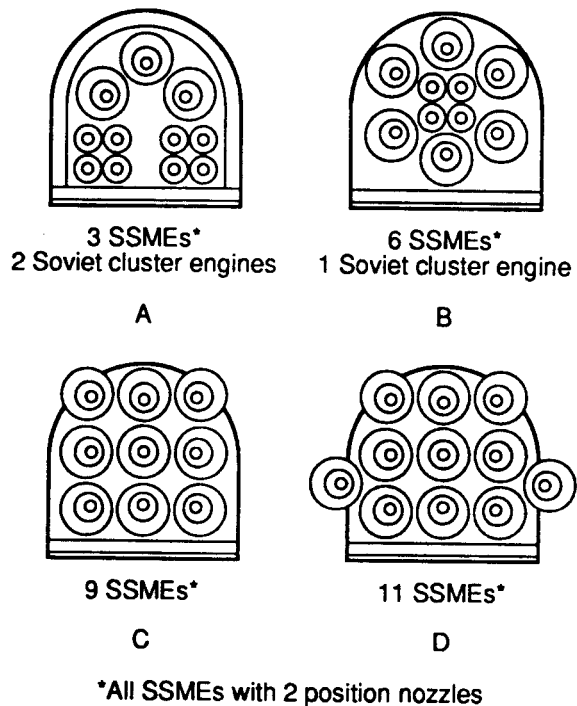


Figure 2. Engine installation patterns.

Table III. Engine Mass Estimating Relationships

Propulsion Configuration <sup>1</sup>	We/TSLs <sup>2</sup>	TVAC/TSLs <sup>3</sup>
3-2 (E = 40/120)	0.0164	1.1070
3-2 (E = 77.5)	0.0162	1.2732
6-1	0.0181	1.1729
9, 10, or 11 SSMEs	0.0187	1.2277

Notes:

1. First numeral refers to number of SSMEs; second numerals to number of RD-170s.
2. Ratio of weight of propulsion system to sealevel thrust.
3. Ratio of vacuum thrust to sea level thrust for the fixed nozzle.

### METHOD OF ANALYSIS

The mass properties of the vehicle were estimated using a program entitled WAMI. This program requires relatively few inputs. The central theme of the program is the use of a non-dimensionalized vehicle length in the mass estimating equations. The non-dimensionalized length, raised to various integer and fractional powers, is used along with derived constants to obtain structural, thermal protection, tank, and propellant weights. The assignment of an actual length of appropriate units is part of the sizing process.

The vehicle length parameter is also used in the determination of overall vehicle c.g. and moments of inertias. In these latter cases subsystem c.g.s and radii of gyration are input as fractions of vehicle length. In some cases, particularly for structure, the values must be calculated outside of the program. For the subsystems, values are assigned from inspection of an inboard profile of the vehicle. These approximations are being accepted as part of the process in reducing input time. The propellant loadings are also part of the input. These values are based on required orbital altitude, inclination, engine performance, and ascent trajectory, and are obtained from the Program to Optimize Simulation Trajectories (POST) (Ref. 4). The assumptions input into the WAMI sizing for the various vehicles are listed in Table IV.

There are, of course, many subsystems for which mass is not dependent upon vehicle size and the vehicle length factor does not appear. In the power system, for example, its weight is assumed to depend entirely upon average and peak power demands. In personnel provisions, the total of the seat weights in each category is dictated by the number of crew and mission specialists or passengers. For the heat rejection system, its size is dependent upon the average power demand, number of crew, and mission lapsed time.

Table IV. Basis for Weights Estimations

**Mission and Vehicle**

- Crew of 2
- Crew cabin volume — 140 ft<sup>3</sup>
- Mission duration — 72 hr
- Payload delivery 28.5° inclination to 160 nmi circular orbit
- Design return payload = 32,000 lb
- Payload accommodation = 1000 lb
- Payload bay volume = 10,000 ft<sup>3</sup>
- On-orbit maneuver  $\Delta V = 933$  ft/sec
- Reaction control  $\Delta V$  equivalent = 80 ft/sec

**Structure and Thermal Protection**

- Wings and body structure are fabricated from composites such as graphite polyimide, graphite epoxy, Kevlar, aluminum-lithium, and Boron aluminum
- Main propellant tanks are fabricated using an aluminum liner overlayed with Nomex honey – comb and overwrapped with a high-strength graphite-epoxy composite
- Thermal protection used is a direct-bond-high-temperature shuttle tile on the windward side of the vehicle with blanket insulation on the leeward side.
- Weight growth margin = 7 percent

For the first series of runs, gross liftoff weight (GLOW) was fixed at approximately 3.6 million pounds; the approximate GLOW for the 1980 paper of reference 1. Each POST and WAMI run had to reflect the constant GLOW of 3,628,604 lb and the weight actually required for each engine and feed system. The baseline combination of 3 SSMEs and 2 RD-170s gives approximately a 30-70 thrust split; 70 percent of the thrust being supplied by the hydrocarbon engines. The known optimum for a dual-fuel system based on studies in references 6 and 7 dictated the numerical selection of 3 SSMEs and 2 RD-170s as being the nearest approach to an optimum dry weight using integral numbers of engines.

In running the WAMI program for fixed engines, the payload is the residual weight at insertion after all subsystems have been sized except engines. Using this procedure, the POST program was run for various propellant mass splits; the propellant mass splits dictate the operating time for the RD-170 and SSME engines. The optimal payload and optimal payload-to-dry weight values were obtained. Having obtained the optimum payload-to-dry weight for operating time of the RD-170 engines for a fixed GLOW, the GLOW was allowed to vary in order to find the optimum GLOW for the optimum operating time of the RD-170 engines (Fig. 5). Additional runs were made for 9, 10, and 11 SSME engines at a fixed GLOW (and for the case of 1 RD-170 and 6 SSMEs).

## **PROGRAM RESULTS**

The combination of 3 SSMEs and 2 RD-170s showed payloads approaching 65,000 lb as compared to 68,000 lb for a combination of 6 SSMEs and 1 RD-170 (Fig. 3). Both runs were made at a fixed GLOW of 3.6 Mlb. Optimums for both payload weight and payload-to-dry weight ratios occurred between 160- and 180-seconds of operation on the RD-170 engines. The payload-to-dry weight ratio for the 3-2 combination of engines was slightly over 0.22 making this configuration the least costly for the assumption that cost is directly proportional to dry weight. The 6-1 combination showed a payload-to dry of 0.20 (Fig. 4).

When GLOW was allowed to vary at a fixed 180-second operating time on the RD-170s, the optimum payload was found to be at a GLOW of slightly over 3.8 million lb (Fig. 5) versus 3.6 million pounds used for the previous runs (Fig. 4). For the 3-2 combination and increased GLOW, the maximum payload increased from 65 to over 70 klb (compare the bottom curve in figure 3 with the top curve in figure 5). Vehicle length was 203 feet compared to 197 feet from reference 1. The increased length is, in part, due to optimization at a higher GLOW at a higher propellant loading.

When the SSME engines with 40/120 nozzles were replaced with the current 77.5 expansion ratio engines in the 3-2 combination, the payload delivered dropped from slightly over 70 klb to 60 klb suggesting that the added weight of 523 lb for each dual position nozzle on the SSMEs was well worth the weight penalty.

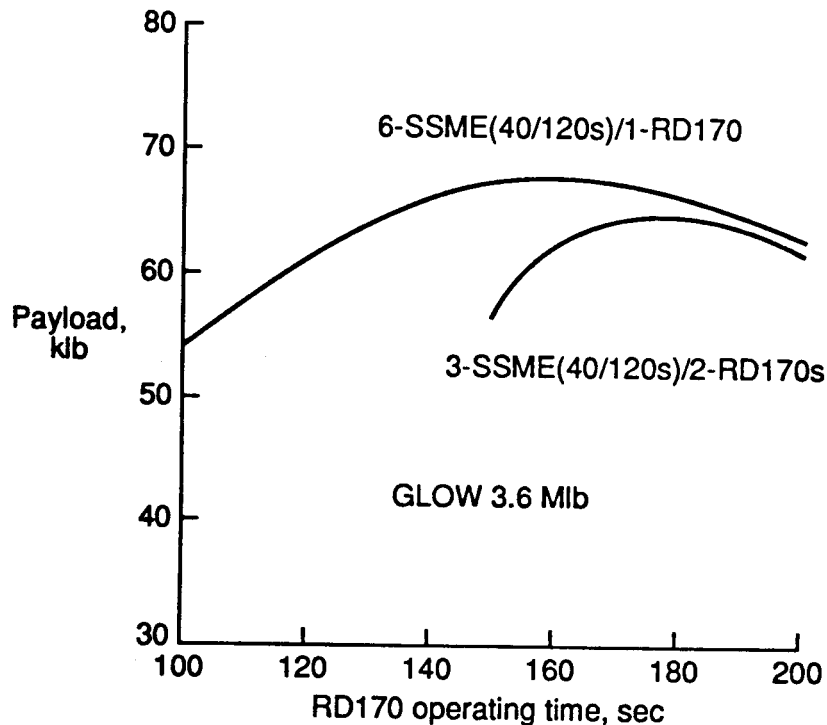


Figure 3. Effect of RD-170 operating time on payload.

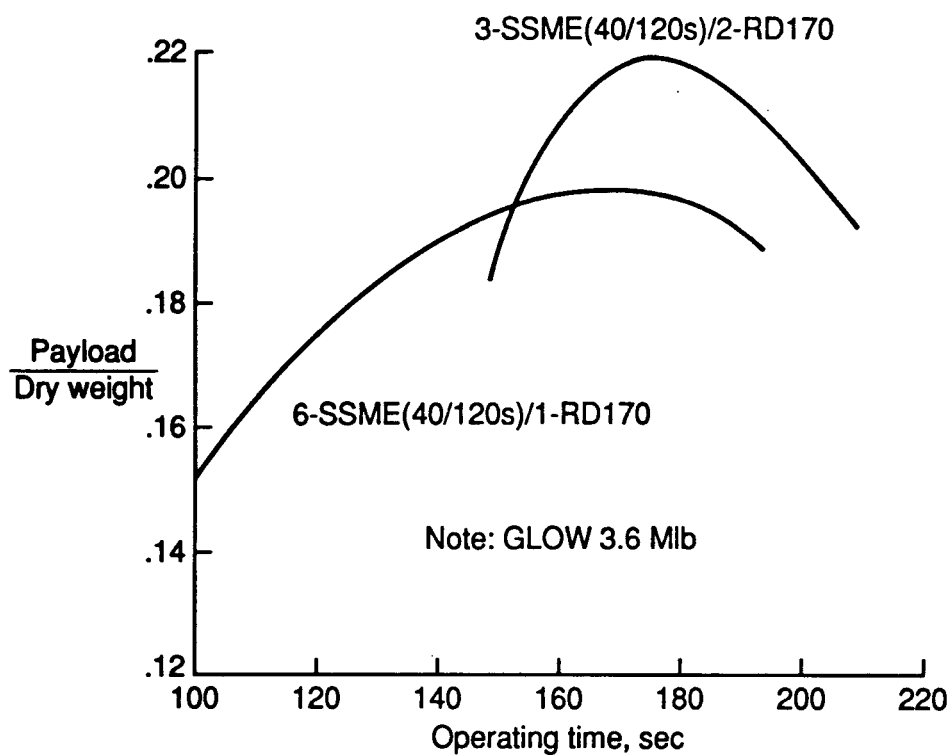


Figure 4. Payload-to-dry weight versus RD-170 operating time.

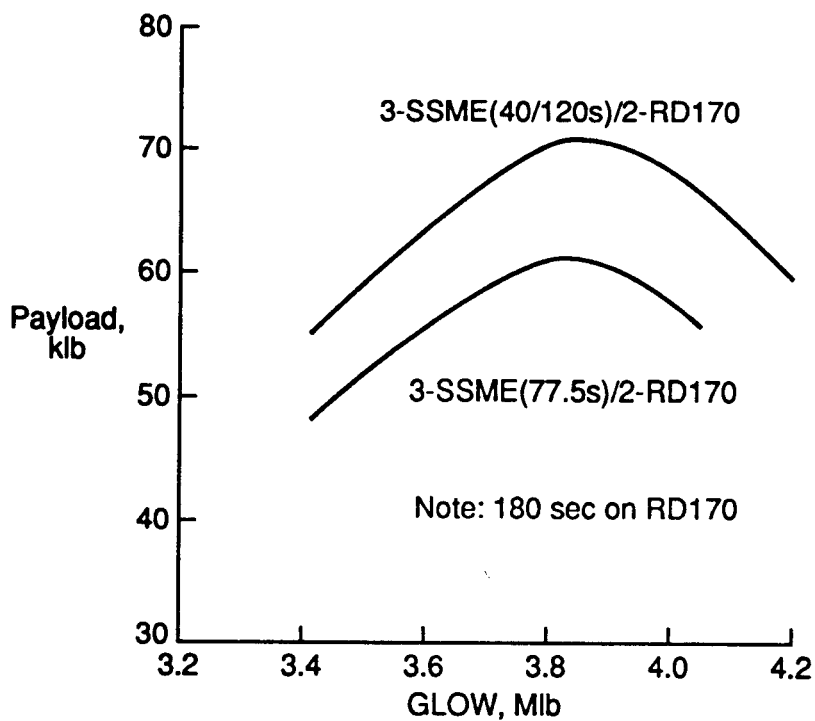


Figure 5. Effect of gross liftoff weight on payload.



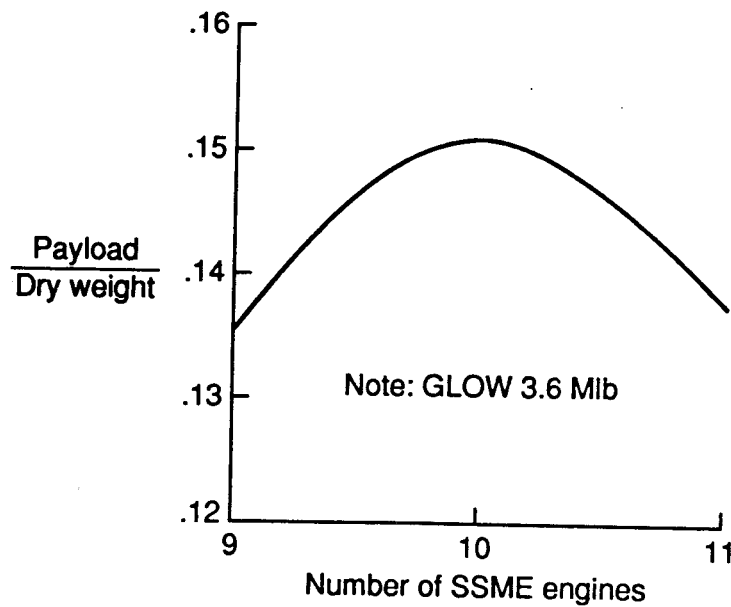


Figure 6. Effect of number of SSME engines on payload-to-dry.

Test runs of all SSME vehicle configurations at 9, 10, and 11 engines showed that the payload-to-dry is optimal for the 10 engine system (Fig. 6). However, this new vehicle length of 224.5 (Fig. 7) represents a 51-percent increase in vehicle volume over the dual-fuel system at 203 ft (Fig. 8). Maximum payload-to-dry weight ratio was 0.15 compared to the 0.22 for the dual-fuel vehicle with 3-2 combination.

Several runs were attempted with an all-RD-170 vehicle but the mass properties program would not converge. Convergence was accomplished only when the engine weights were artificially reduced by 75 percent.

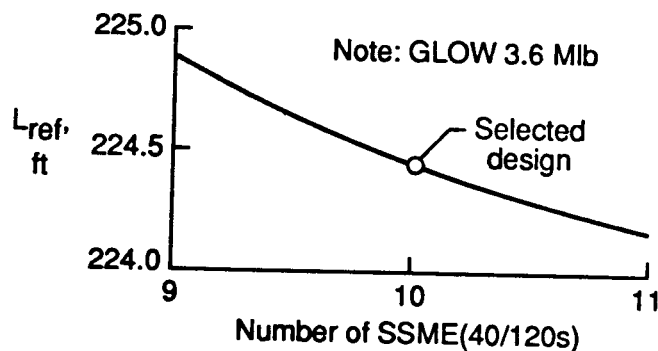


Figure 7. Vehicle length versus no. of SSME engines.

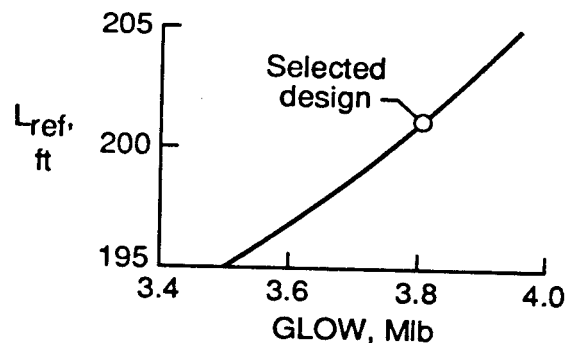


Figure 8. Length of dual-fuel vehicle versus gross.

## IMPACT OF PROPULSION SYSTEM CHANGE ON VEHICLE

The original vehicle of reference 1 was configured for a nominal center of gravity (c.g.) location at entry and landing of 72 percent. All wind tunnel tests were run at the 72 percent value (Refs. 8, 9, and 10). However, results from the mass properties program indicate that c.g. locations are in the range of 74 to 75 percent when using the heavier RD-170 engines. In order to obtain the best compromise for c.g. location for conditions of payload-in and payload-out at entry, it was necessary to move the crew compartment from the mid-body station, where the cargo compartment is located, to the ogive section ahead of the hydrogen tank (Fig. 9). In addition, it was necessary to re-configure the crew compartment and increase its size to a crew complement of 8 instead of 2 in order to have the least excursion in c.g. location from the 72 percent value for payload-in and payload-out cases (bottom curve in figure 10).

The crew compartment, in the form of a ballistic capsule, is over 20 ft long and 18 ft in diameter and is estimated to weigh 13000 lb (Fig. 11). Assuming the difference in weight of the 8-man and 2-man capsules is 8000 lb, then the optimal payload indicated by the top curve in Figure 5 would be reduced from 70 klb to approximately 62 klb.

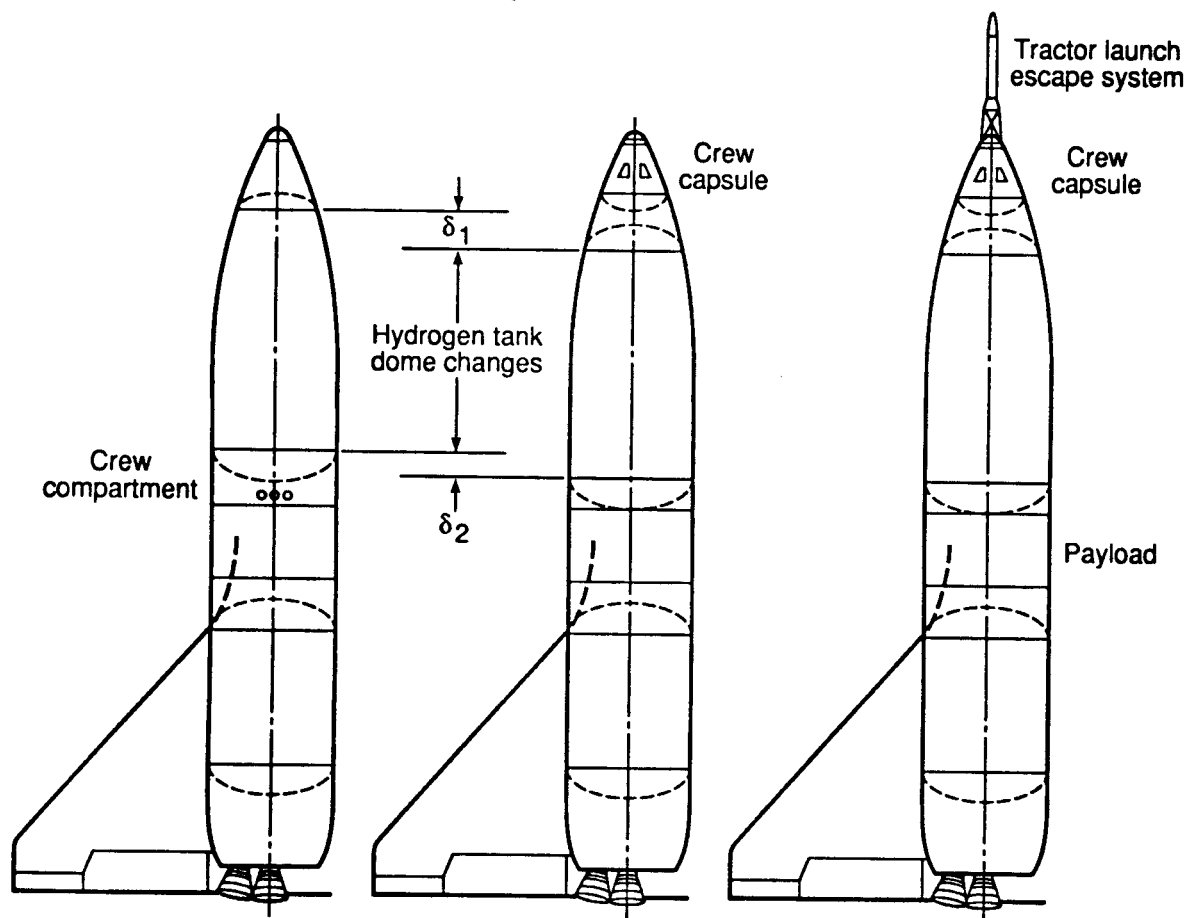


Figure 9. Crew and escape system options.

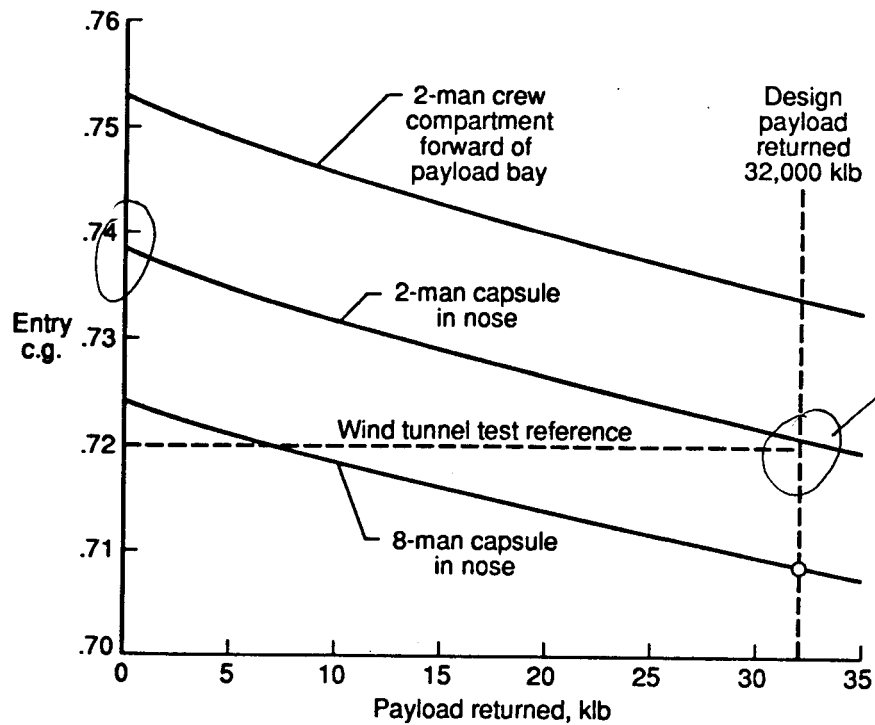


Figure 10. Center of gravity versus payload returned.

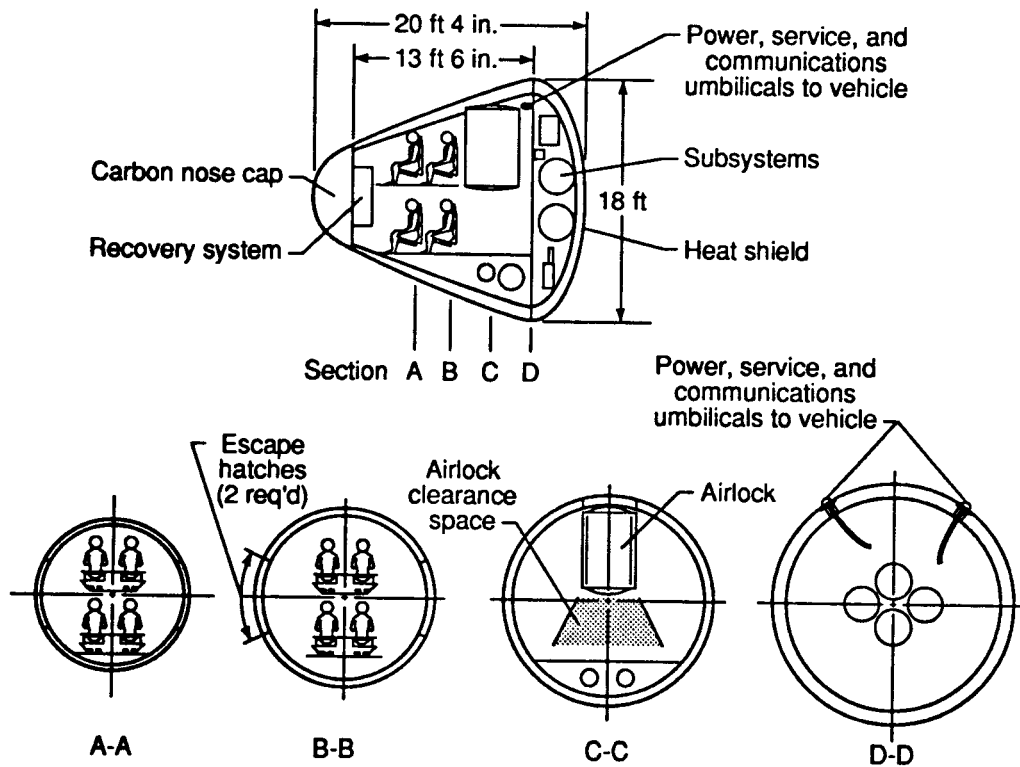


Figure 11. Crew capsule cross sections.

## OPERATIONS AND MANUFACTURING

The dual-fuel single-stage vehicle would be much simpler to operate than the current Shuttle. There are no solids or hypergols, no cross-feed, and only one vehicle to service. Because there is only one stage to manufacture, only one set of tooling is required.

If the body diameter is restricted to 32.8 ft, this matches the diameter of the Saturn 1st stage that was manufactured in the Michoud facility, in New Orleans, Louisiana; the same facility now used to manufacture the 28.6 ft diameter external tanks for the Shuttle. For any diameter larger than 32.8 ft, a determination would have to be made as to overhead clearance when the cradle and vehicle height are both considered.

It is probably not practical to carry the dual-fueled single stage on top of a 747 because of its size (Fig.12). For this reason, two bolt-on 747 engines are proposed (Fig.13). The bolt-on engines would also be used in flight-test. To simulate "dead stick" landing without engines, the engines would be operated at zero net thrust to simulate the drag of a vehicle with no engines. Mechanical, electrical, and hydraulic connections are required at the engine pylon-to-vehicle interface.

One advantage of the large size in Earth-to-orbit transport vehicles, is the fabrication process, in that the metal gauges are greater, making the components easier to join. When a launch system is driven to two stages more of the metallic components, especially tankage, become minimum gauge and inspection costs go up, while the life of the tanks goes down. It is also easier to integrate a large payload into a large vehicle in the design process and entry c.g. location excursion is less for a large vehicle than a small one with changes in payload.

A disadvantage of a single stage over a multistage vehicle is that crew and payload compartment tend to be higher off the pad. For the 203 ft single stage with SSME and Soviet engines in the 3-2 combination, the difference between crew capsule height and Shuttle cabin off the pad is estimated to be 86 ft. From centroid-to-centroid of volume, the payload bay compartment is 23 ft higher off the pad in the single stage (Fig. 14); however, the payload compartment deck is flat and is only 15 ft high along the vehicle's major axis versus 60 ft for the current Shuttle.

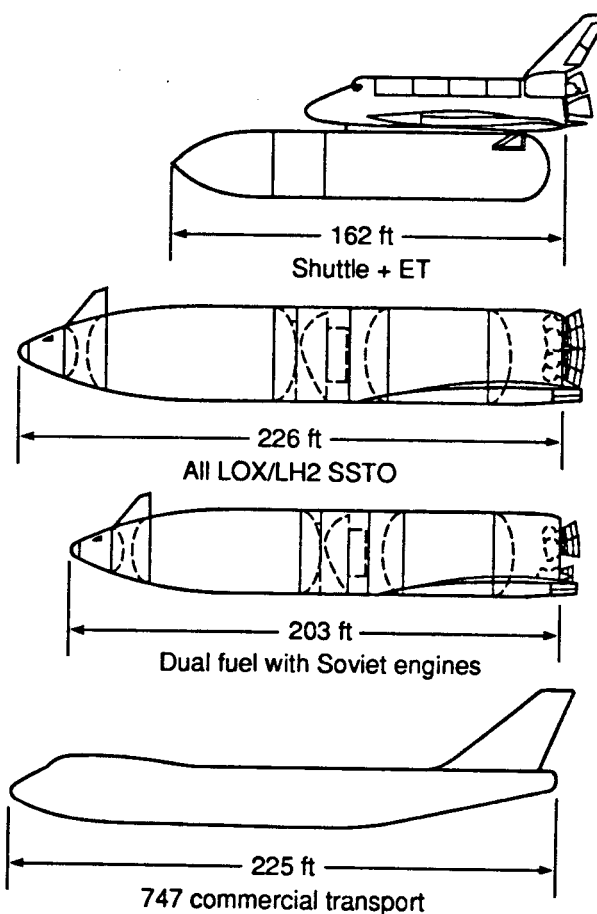


Figure 12. Vehicle size comparisons.

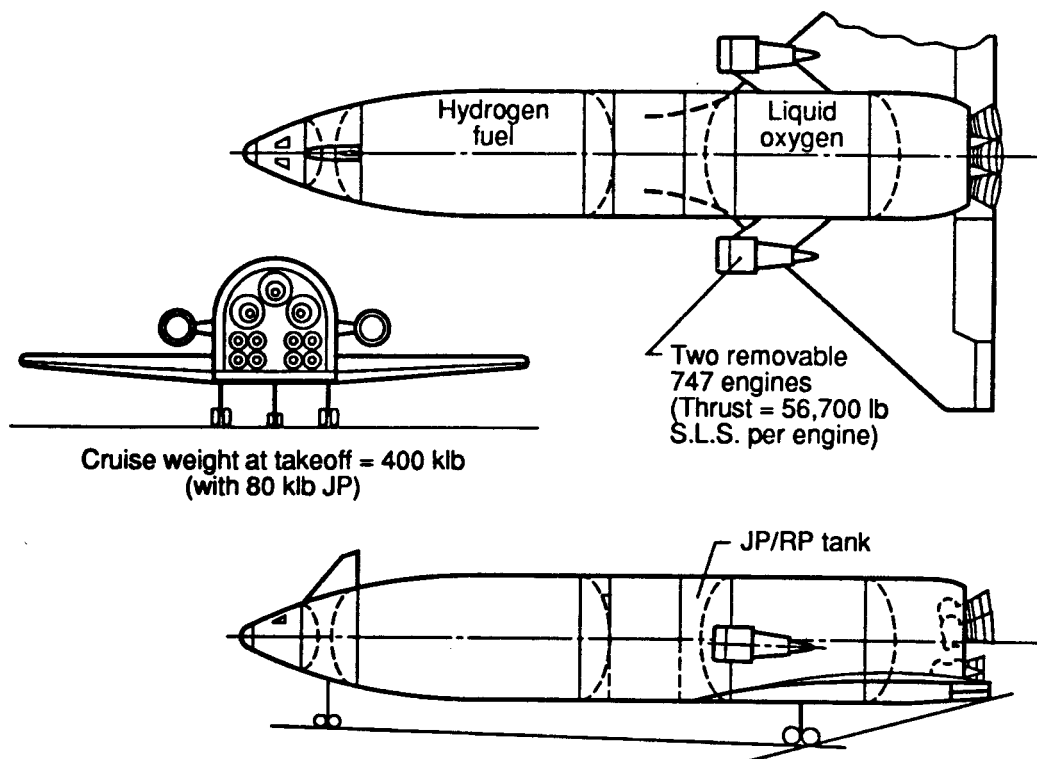


Figure 13. Dual-fuel vehicle cruise system.

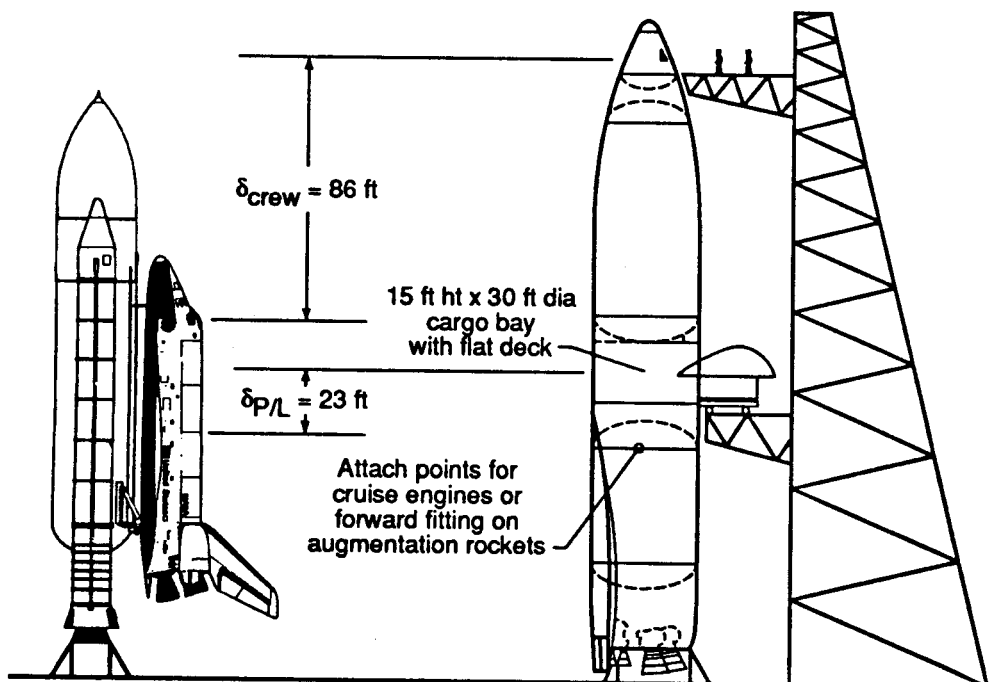


Figure 14. Comparison of dual-fuel vehicle with Shuttle.

## **SUMMARY REMARKS**

The substitution of existing Soviet RD-170 engines for those identified in Reference 1 resulted in:

- 1) an increase in vehicle dry weight because of heavier engines.
- 2) an increase in the crew compartment weight and location to balance the added weight of engines at the rear of the vehicle.

The weight increase of propulsion on the vehicle is attributed to the following:

- 1) The Soviet engine is existing hardware while the weight of the previously used engine is a projected weight.
- 2) The Soviet engine represents older technology.
- 3) The allowance used for pressurization and feed in the earlier study of Reference 1 may not have been adequate.

## **REFERENCES**

1. MacConochie, Ian O.; and Klich, Phillip J.: "Technologies Involved in Configuring An Advanced Earth-to-Orbit Transport For Low Structural Mass." SAWE Paper No. 1380. Presented at the 39th SAWE Conference, May 12-14, 1980.
2. Anon: "Soviets Reveal Energia Details." Flight International Sept. 17, 1988, p. 65.
3. Luscher, W. P.; and Mellish, J. A.: "Advanced High Pressure Engine Study For Mixed-Mode Vehicle Applications", NASA CR-135141. Final Report prepared by The Aerojet Liquid Rocket Co. for Lewis Research Center under Contract NAS 3-19727.
4. Brauer, G. L.; Cornick, D. E.; and Stevenson, R.: "Capabilities and Applications of the Program to Optimize Simulated Trajectories (POST) - Program Summary Document." NASA CR-2770, Feb. 1977.
5. MacConochie, Ian O.; et al "Shuttle II: Subsystem Weights Program Development." SAWE Paper No. 1764. Presented at the 46th Annual Conference, Seattle, WA, May 1987.
6. Salkeld, Robert: "Mixed-Mode Propulsion of the Space Shuttle" Astronautics and Aeronautics, Vol. 9, No. 8, Aug. 1971.

7. Rehder, J. J.: "Effect of Propulsion System Characteristics on Ascent Performance of Dual-Fueled Single-Stage Earth-to-Orbit Transports." NASA TP 1115, Dec. 1977.
8. Wells, William L.; et al: "Heating Rate Distributions at Mach 10 in a Circular Body Earth-to-Orbit Transport Vehicle." AIAA paper No. 85-0974. Presented at the AIAA 20th Thermophysics Conference, Williamsburg, VA, 1985.
9. Lepsch, R. A.; and MacConochie, I. O.: "Subsonic Aerodynamic Characteristics of a Circular Body Transport." AIAA paper No. 86-1801-CP. Presented at the AIAA 4th Applied Aerodynamics Conference, San Diego, California, June 9-11, 1986.
10. MacConochie, I. O. et al.: "Advantages of a Round Body Shuttle," Aerospace America, July 1989, pp. 44-46.

## Biographies

Ian O. MacConochie received a B.M.E. in Mechanical Engineering from the University of Virginia in 1950 and a D.I.C. (Diploma Imperial College, London) in 1958 specializing in gas turbines and lubrication. From 1962 to 1989 he was employed by NASA Langley Research Center, first in structures, and then in engineering. Prior to his employment at NASA, and during his early engineering career, he taught mechanical engineering at Duke University and the University of South Carolina. Presently he is an employee of the Lockheed Engineering and Sciences Company, supporting NASA in their studies of advanced space transportation systems.

Douglas O. Stanley received a B.S. in Mathematics from Baylor University in 1986 and an M.S. in Astronautical Engineering from George Washington University in 1988. He is currently employed in the Space Systems Division at NASA Langley Research Center where he is responsible for performance, propulsion, and configuration analysis in support of a variety of advanced space transportation studies. He is currently serving as NASA Langley's representative on the Government Evaluation Team for the ongoing Single-Stage-to-Orbit Technology Studies through the Strategic Defense Initiative Office.